Copy No.

~~CONFIDENTIAL~~

RM No. L6I10

P-51D

RESEARCH MEMORANDUM

EFFECT OF MACH NUMBER ON THE MAXIMUM LIFT AND BUFFETING BOUNDARY DETERMINED IN FLIGHT ON A NORTH AMERICAN P-51D AIRPLANE

By

John P. Mayer

Langley Memorial Aeronautical Laboratory
Langley Field, Va.

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

WASHINGTON

June 12, 1947

~~CONFIDENTIAL~~

NACA RM No. L6I10 [redacted]

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

RESEARCH MEMORANDUM

EFFECT OF MACH NUMBER ON THE MAXIMUM LIFT AND BUFFETING BOUNDARY DETERMINED IN FLIGHT ON A NORTH AMERICAN P-51D AIRPLANE

By John P. Mayer

SUMMARY

Flight tests were conducted on a North American P-51D airplane to establish the maximum lift coefficient and the buffeting boundary line as a function of Mach number. Abrupt stalls were made at Mach numbers from 0.21 to 0.63 and gradual stalls were made at Mach numbers from 0.41 to 0.65. The buffeting boundary was determined in abrupt pull-ups through a Mach number range from 0.21 to 0.80.

The results indicate that the maximum lift coefficient and the buffeting boundary line as established in abrupt pull-ups were very much affected by Mach number and that Reynolds number had no apparent effect on maximum lift coefficient in abrupt pull-ups within the limits of the test data.

Up to a Mach number of 0.64 the buffeting boundary was defined by the actual limit maximum lift coefficient attainable with the P-51D airplane in abrupt pull-ups. Above a Mach number of 0.64 the buffeting boundary dropped sharply and was below the actual maximum lift coefficient of the airplane.

A comparison between the buffeting boundary found in the flight tests and a calculated wing buffeting boundary shows good agreement up to a Mach number of 0.42 with a lesser degree of agreement at higher Mach numbers.

The gradual stalls of the airplane indicated that the maximum lift coefficient was affected by Mach number in a manner similar to that for the abrupt stalls.

[redacted]

INTRODUCTION

Although there is considerable wind-tunnel material available on the variation of the maximum lift coefficient with such factors as Reynolds number and airfoil shape, there is less known about the effects of either Mach number or rate of change of angle of attack on maximum lift coefficient, both of which are becoming increasingly important. Also, the occurrence of buffeting at high Mach numbers and lift coefficients lower than the maximum lift coefficient has imposed an effective limit in lift on the airplane beyond which pilots have seldom ventured. Relatively few data exist on this latter phase of the problem and little is known concerning the prediction of this limit.

In the course of a high-speed dive test program on a P-51D airplane at Langley Memorial Aeronautical Laboratory of the National Advisory Committee for Aeronautics at Langley Field, Virginia, some data on the variation of maximum lift coefficient and buffeting lift coefficient with Mach number were obtained. This report presents the results of these tests. The true maximum lift coefficients were measured in abrupt and gradual stalls up to a Mach number of 0.63, whereas the buffeting boundary was established up to a Mach number of 0.80.

The present results extend the available flight data on abrupt stalls of airplanes with low drag wings (results of Ames Laboratory tests of the Bell P-63A-6 airplane) from a Mach number of 0.44 to 0.63. Although the tests of the P-51D airplane did not extend the Mach number range of other investigations with regard to the buffeting boundary (references 1 and 2), the instrumentation of the airplane was such that the buffeting boundary could be quite accurately determined. In addition, since tail loads were measured on the P-51D airplane, wing lift coefficients as well as airplane lift coefficients were evaluated.

APPARATUS

Description of Airplane

The airplane used in the tests was a North American P-51D, reinforced structurally to withstand the high loads expected in a high-speed dive program in progress at Langley Laboratory. Figure 1 shows a side view of the airplane used in the flight tests.

The general specifications of the airplane as flown are as follows:

Airplane	North American P-51D Army Air Forces No. 44-13257
Engine	Packard built Rolls Royce V-1650-7 12 cylinder
Propeller	Hamilton Standard 4-blade hydromatic
Diameter, feet	11.17
Blade number	K6523A-24
Weight at take off, pounds	8850
Center-of-gravity position (at take off), percent M.A.C.	25.1
Wing:	
Span, feet	37.03
Area, square feet	240.1
Dihedral (at 25 percent chord), degrees	5
Sweepback (leading edge), degrees	3.6
M.A.C., inches	79.6
Airfoil	NAA-NACA low drag
Horizontal tail:	
Area, square feet	28.0
Incidence, degrees	1

Instrumentation

Airspeed, pressure altitude, and airplane normal acceleration were measured as functions of time with standard NACA recording instruments. The tail normal acceleration was measured with a Statham accelerometer in connection with a Miller 15-element recording oscillograph. Loads on the wing and tail were found by using strain-gage measurements recorded on the Miller oscillograph.

The airspeed head was mounted on a boom extending 1.2 local chord lengths ahead of the leading edge of the wing and located near the right wing tip of the airplane. The airspeed-altitude recorder was located in the right wing so as to minimize lag effects. This airspeed system was calibrated for position error, due to lift coefficient and Mach number effects, up to a Mach number of 0.78.

The strain-gage installation on the airplane was calibrated periodically by applying known loads to the wing and tail of the airplane.

FLIGHT-TEST PROCEDURE

All flight tests were made with the airplane in the clean condition and with power on.

Abrupt stalls were made at pressure altitudes of 10,000, 20,000, and 30,000 feet at Mach numbers from 0.21 to 0.63. In these stalls the airplane was pulled up as abruptly as possible, the degree of abruptness depending upon the inertia, control power, and stability of the airplane as flown. A series of gradual stalls was also made in turns at 30,000-foot-pressure altitude at Mach numbers from 0.41 to 0.65.

In the pull-ups within the Mach number range from 0.64 to 0.80, maximum lift coefficients were not reached because of buffeting. In this range the airplane was pulled through the buffeting boundary until the vibration of the airplane became objectionable to the pilot at which point recovery from the pull-up was made and buffeting stopped. The pull-ups through the buffeting boundary were made somewhat more slowly than the low-speed pull-ups.

METHOD

In order to illustrate the definitions and methods employed in evaluating results, three typical load-factor time-history diagrams obtained in abrupt pull-ups are shown in figure 2. Point A in each of the diagrams represents the point where buffeting started; B, the point of peak mean load factor; and C, the point where buffeting stopped. In figures 2(a) and 2(b) the first two points coincide, while in figure 2(c) the peak load factor occurs after buffeting starts and between points A and C.

From the data of the type shown in figure 2 the airplane and wing lift coefficients were evaluated for a number of runs at the points where buffeting started and stopped as well as at maximum lift. In computing lift coefficients the lift was assumed to be equal to the normal force, and fuselage and propeller normal loads were neglected. The equations used in determining lift coefficients were:

$$C_{L_A} = \frac{nW}{qS}$$

$$C_{L_W} = \frac{nW - L_T}{qS}$$

where

C_{L_A} airplane lift coefficient

C_{L_W} wing lift coefficient

n normal load factor (measured perpendicular to airplane thrust line)

q dynamic pressure, pounds per square foot

S wing area, square feet

W airplane weight, pounds

L_T horizontal tail load, pounds, as determined from the strain gages and accelerometer records

Since the tests of the P-51D as well as other investigations (references 3 and 4 and results of Ames Laboratory tests of the Bell P-63A-6 airplane) indicate that the maximum lift coefficient depends on the pitching angular velocity, the maximum lift coefficients obtained in the abrupt pull-ups were plotted versus the angle of pitch per chord length traveled. This parameter is

$$\frac{C}{V}\frac{d\alpha}{dt}$$

where

C mean aerodynamic chord, feet

V airspeed, feet per second

$\frac{d\alpha}{dt}$ time rate of change of angle of attack, radians per second

The rate of change of angle of attack $d\alpha/dt$ was in turn determined from the measured rate of change of load factor with time and the equation

$$\frac{d\alpha}{dt} = \frac{W/S}{dC_L/d\alpha} \frac{dn/dt}{q}$$

where

$dC_L/d\alpha$ slope of lift curve, per radian

dn/dt time rate of change of load factor

The slope of the lift curve $dC_L/d\alpha$ at the various values of Mach number was obtained from unpublished data from wind-tunnel tests made at Ames of the XP-51 airplane. The slope of the load-factor time diagram was taken at the time corresponding to 6 chord lengths before the maximum acceleration was reached. This corresponds approximately to the time the lift coefficient lags the angle of attack when the angle is changing rapidly.

ACCURACY

The estimated accuracy in the determination of the pertinent results is as follows: C_{L_A} or C_{L_W}, < 3 percent, M, $\pm$0.01, and $\frac{C}{V}\frac{d\alpha}{dt}$, $\pm$15 percent.

These probable errors arise principally from errors in the measurement of dynamic pressure, pressure altitude, load factor, and, in the case of lift coefficients, in the assumption that the lift was equal to the normal force. In the determination of $\frac{C}{V}\frac{d\alpha}{dt}$, however, the listed error is attributed to (1) the necessity of using wind-tunnel data from tests of a model of the XP-51 for lift-curve slope, (2) the somewhat arbitrary selection of the point at which the slopes were read, and (3) graphical errors in the differentiation process.

RESULTS AND DISCUSSION

The Effects of Mach and Reynolds Number on the Maximum Lift Coefficient

The results of a number of abrupt pull-ups to the maximum lift coefficient for those cases where points A and B coincide (fig. 2) and the results of gradual stalls are presented in figure 3. The results shown indicate that the airplane maximum lift coefficient obtained in the abrupt stalls decreases rapidly as the Mach number increases from 0.21 to 0.48 where a minimum point is reached. The maximum lift coefficient then increases until a secondary peak is reached at a Mach number of 0.56 after which it again begins to decrease rapidly to the limit of the present tests. The secondary peak in the maximum lift coefficient is characteristic of low drag airfoils and is caused by the broadening of the upper surface low-pressure region which offsets the reduction in the negative pressure peak as the Mach number increases. As the Mach number increases further the decrease in the negative pressure peak more than accounts for the broadening upper surface pressure and the maximum lift coefficient again begins to decrease. It can also be seen from figure 3 that altitude, and therefore Reynolds number, has no apparent effect on the maximum lift coefficient obtained in abrupt stalls within the limits of the data obtained. This result has also been shown in reference 5 for the P-47C airplane and in the results of Ames Laboratory tests of the P-63A airplane. In the curve of figure 3 it is also seen that the general trend for the gradual stalls is similar to that for the abrupt stalls with the minimum and peak maximum lift coefficients occurring at similar Mach numbers.

A comparison of the results obtained in the abrupt stalls with similar results obtained with a P-63A airplane (fig. 4) qualitatively indicates the same sort of variation for the two cases. The differences noted between the two cases may be ascribed to the fact that, although both wings are of the low drag type, the sections are dissimilar; those on the P-63 being obtained from the NACA 66 series of airfoils while those of the P-51D are a North American-NACA compromise section. It is to be noted, also, that the abrupt pull-ups for the P-63 were not carried sufficiently far to indicate any minimum point in the C_{L_A} curve.

Comparison between results of gradual stalls of a P-51B (reference 6) and those of the P-51D (fig. 4) show fair agreement throughout. Whatever differences exist may be attributed to the fact that the two airplanes have a slightly different configuration.

Effect of Mach Number on the Buffeting Boundary

Figure 5 is an extension of the results given in figure 3 to include those pull-ups where buffeting prevented the attainment of true maximum lift. (See fig. 2(c).) The pull-up traced out by the curve A, B, C illustrates the manner of variation of lift coefficient with Mach number obtained in a typical high Mach number pull-out.

From a Mach number of 0.21 to 0.64 the buffeting boundary is defined by the actual limit maximum lift coefficient as obtained in abrupt pull-ups of the airplane. Above a Mach number of 0.64, however, the buffeting lift coefficients are below the maximum lift coefficients. It is seen from figure 5 that the lift coefficient at which buffeting either starts or stops decreases rapidly with Mach number and that at a Mach number of about 0.83 buffeting would occur even at zero lift. The implication of the results of figure 5, insofar as they specifically apply to the P-51D airplane, is given in figure 6 where the lift capabilities of the P-51D are shown for several altitudes. The portions of the curves below $M = 0.64$ were established from the solid part of the curve in figure 5 and the portions above $M = 0.64$ were established from the dotted part of the curve. It is seen that at 40,000 feet the airplane would be capable of only the mildest maneuvers and that even at 1 g buffeting would occur at $M = 0.79$.

It may also be seen from figure 5 that the lift coefficients where buffeting starts and stops apparently define a single curve in the region from $M = 0.64$ to $M = 0.80$. In the $C_{L_{max}}$ region (solid curve) the lift coefficient where buffeting stops lies below the point where it initially started. An indication of this result may be obtained from figure 2(b). However, in this range, the lift coefficient where buffeting stopped depended upon the rate of change of angle of attack, and, in general, seemed to be lower than the gradual stall line.

Several papers have presented charts by which the low-speed negative pressure coefficients may be expanded to account for effects of compressibility. In general, such charts when used to expand each pressure point along the airfoil can be made to yield a variation of a critical lift coefficient with Mach number; the word "critical," then, being associated with the attainment of the velocity of sound over some portion of the airfoil. In general, flight observations as well as wind-tunnel experience have not indicated serious effects when the local velocity of sound is first reached. Therefore, curves of critical lift versus M would be expected to lie well below the curve shown in figure 5 and could only serve as a rough guide to the buffeting limit. The charts of reference 5 make possible a prediction of the buffeting limit rather than a critical lift coefficient.

The charts of reference 5 have been applied to expand the theoretical pressure distributions over the mean aerodynamic chord section of the P-51D in order to obtain the variation of the limit lift or buffeting lift coefficient with Mach number. Figure 7 illustrates the agreement between the results calculated in this manner and the experimental results of figure 5. It can be seen that although the computed limit lift curve follows the trend of the measured results it is not as close as would be desired for quantitative purposes.

Effect of Angular Velocity on Maximum Lift Coefficient

Figure 8 shows the results of the effect of rate of change of angle of attack on the maximum lift coefficient for the P-51D airplane, in abrupt stalls, at four mean Mach numbers. The values of the maximum lift coefficient for zero angular velocity were taken from the mean line for the gradual stalls. The lines of constant $C_{L_{max}}$ shown in the figure for the four mean Mach numbers were taken from the mean line through the test points, given in figure 3, for the abrupt stalls.

Figure 8 indicates that throughout the Mach number and $\frac{c}{V}\frac{d\alpha}{dt}$ range covered in the P-51D tests the variation of maximum lift coefficient with the angle of pitch per chord length traveled $\frac{c}{V}\frac{d\alpha}{dt}$ is relatively constant. This is in agreement with results of Ames Laboratory tests of P-63A-6 airplane in which it was shown that the maximum lift coefficient increases almost linearly with angular velocity until a limiting value of the maximum lift coefficient is reached which is unaffected by further increases in angular velocity.

~~CONFIDENTIAL~~

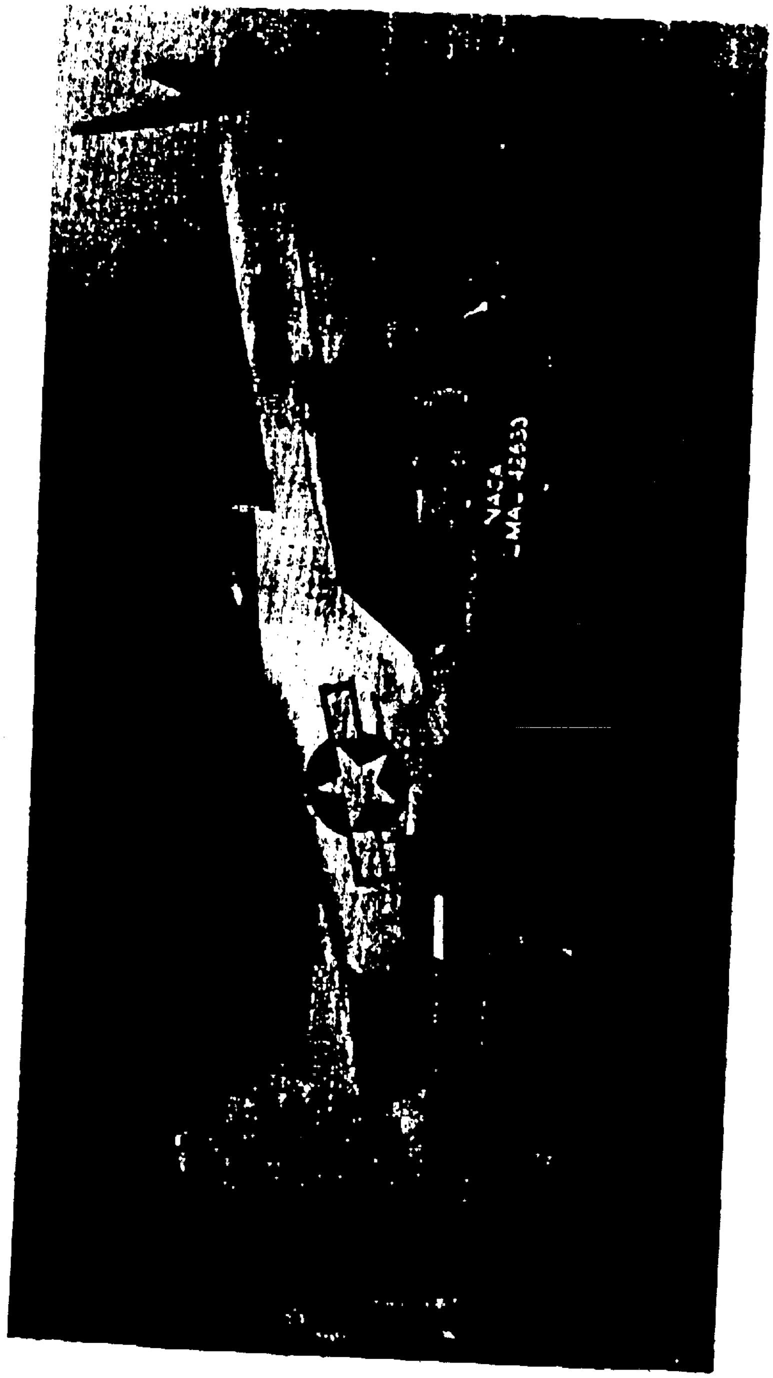


Figure 1.- Side view of P-51D airplane used in tests.

~~CONFIDENTIAL~~

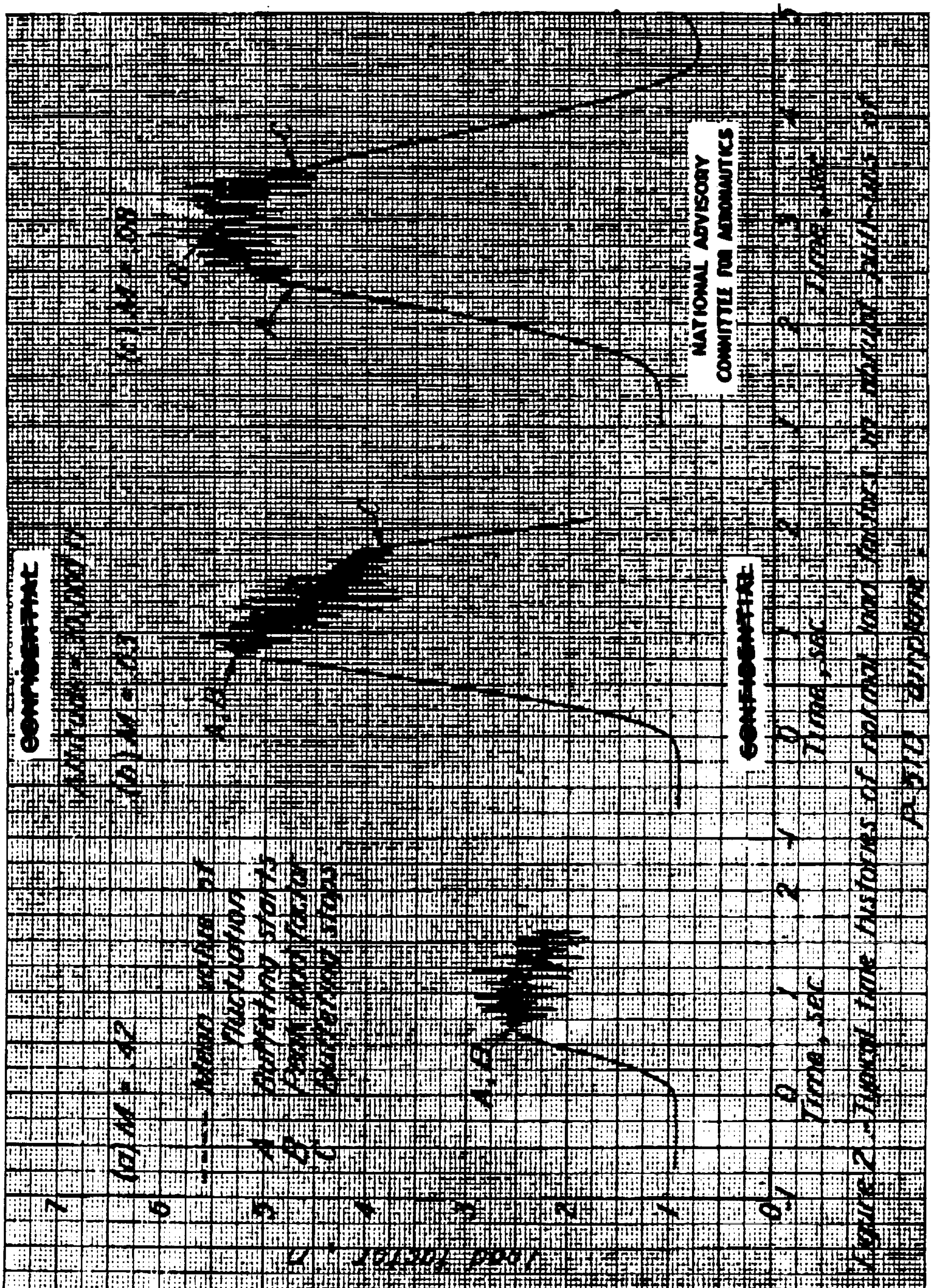


Figure 2.- Typical time histories of normal load factor at maximum lift coefficient. P-51D airplane.

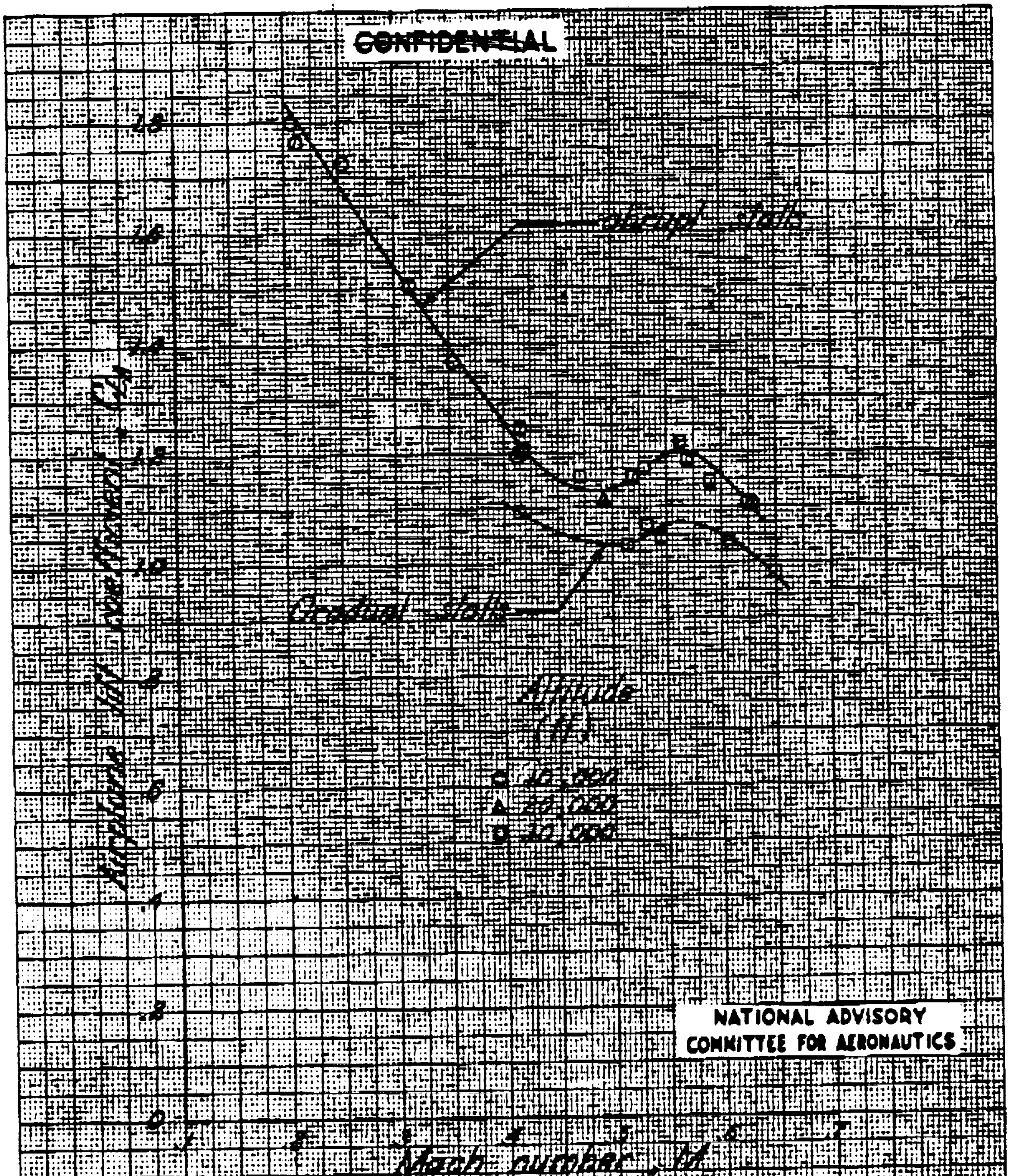


Figure 3.- Maximum lift coefficients obtained in abrupt and gradual stalls for P-51D airplane.

CONFIDENTIAL

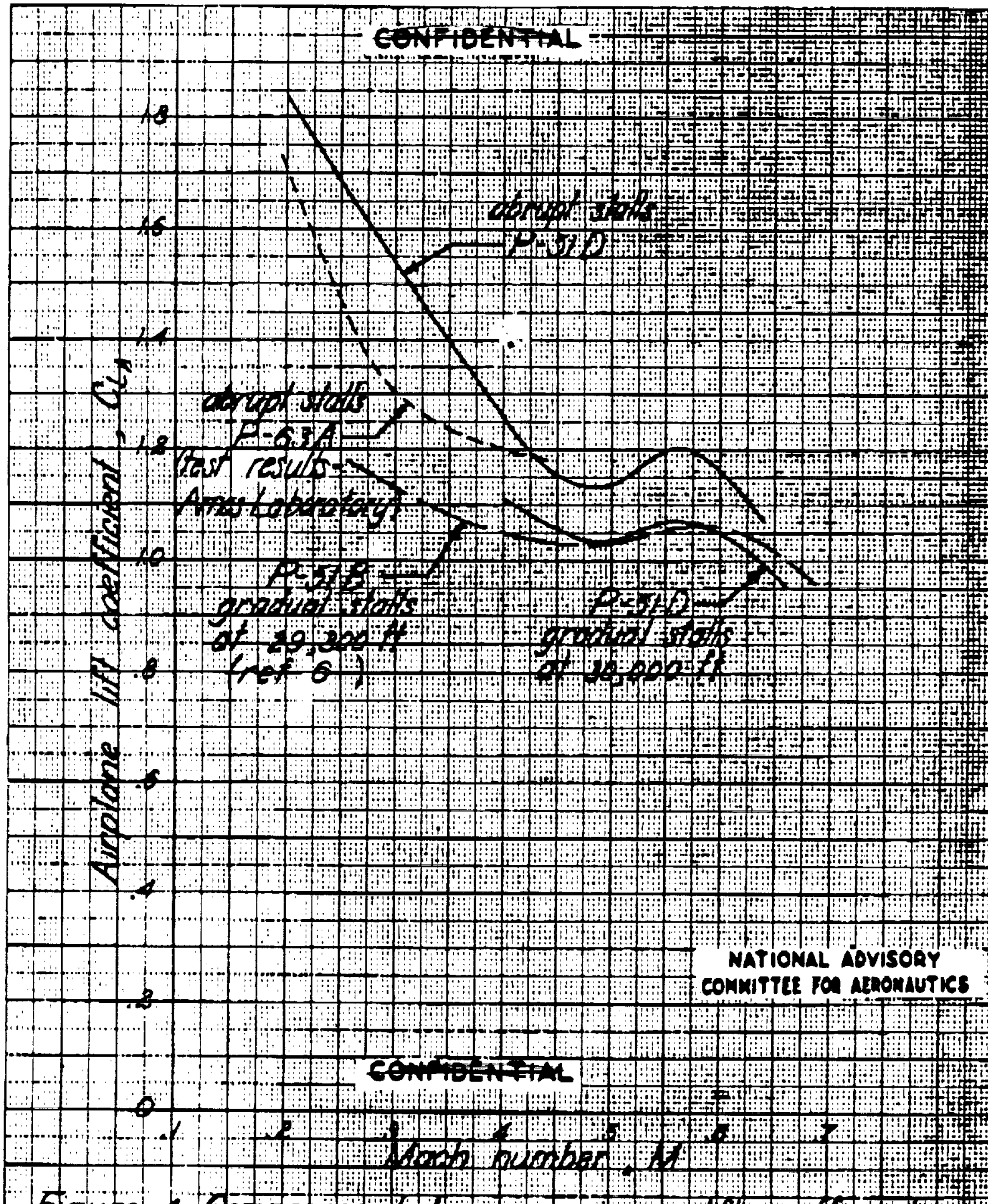
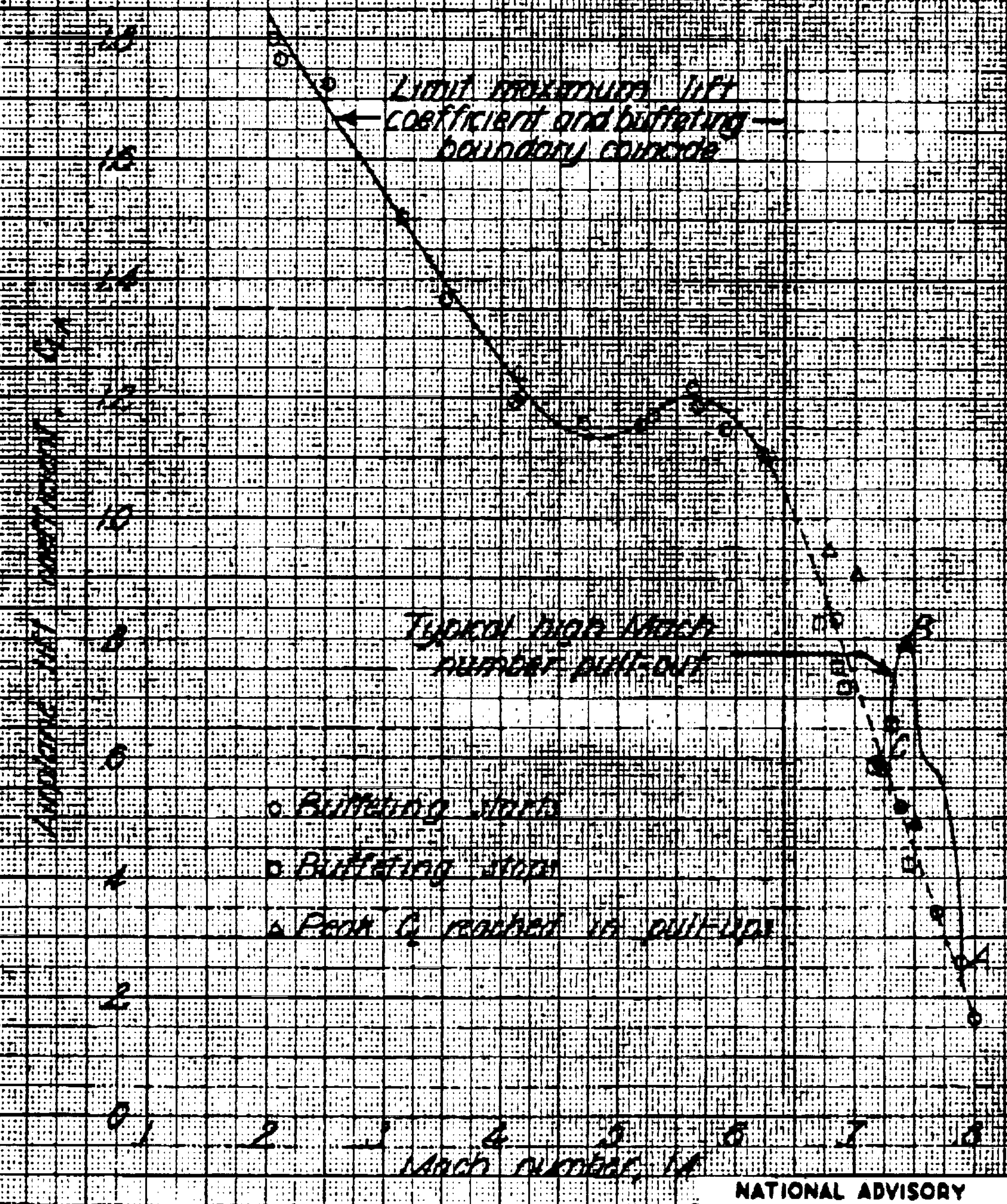


Figure 4.- Comparison between maximum lift coefficients obtained in abrupt stalls of P-51D and P-63A airplanes and those obtained in gradual stalls of P-51D and P-51B airplanes.

CONFIDENTIAL

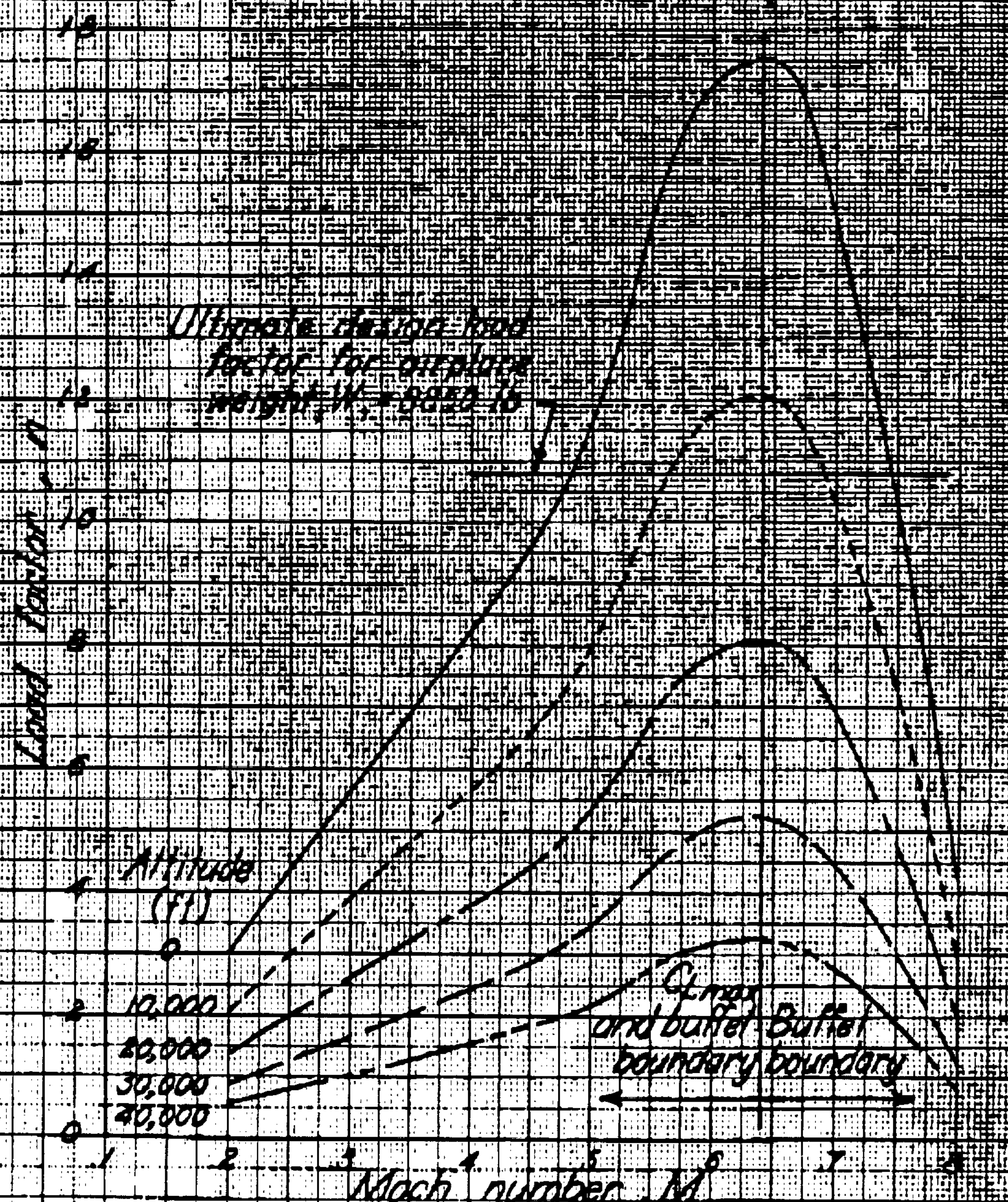


NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

Figure 5.- Buffeting boundary for P-51D airplane.

CONFIDENTIAL

CONFIDENTIAL



CONFIDENTIAL

Figure 6 - Buffeting boundary at various altitudes for P-51D airplane

NATIONAL ADVISORY COMMITTEE FOR AERONAUTICS

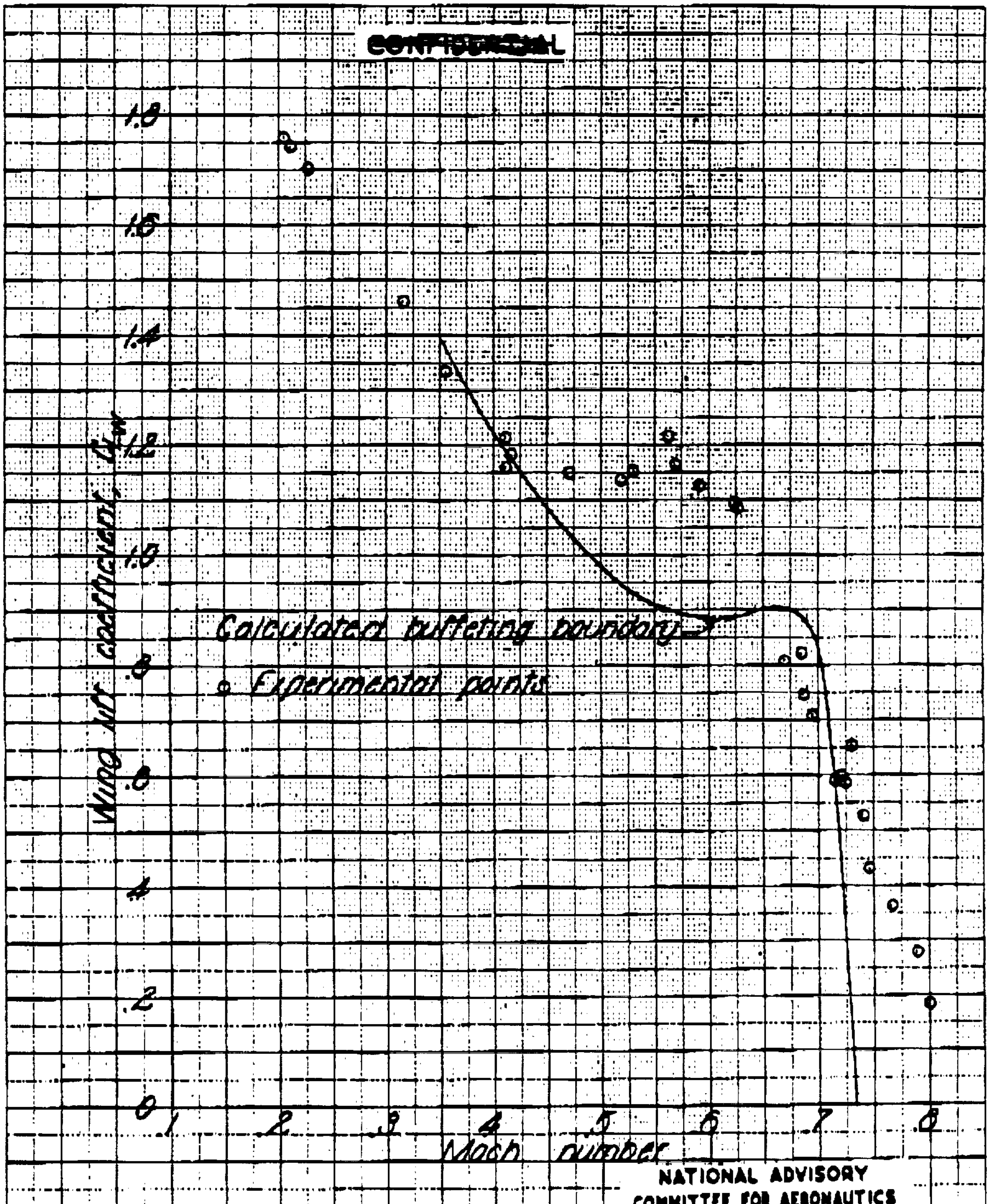


Figure 7 - Comparison between calculated and experimental buffeting boundary for P-51D airplane.

~~CONFIDENTIAL~~

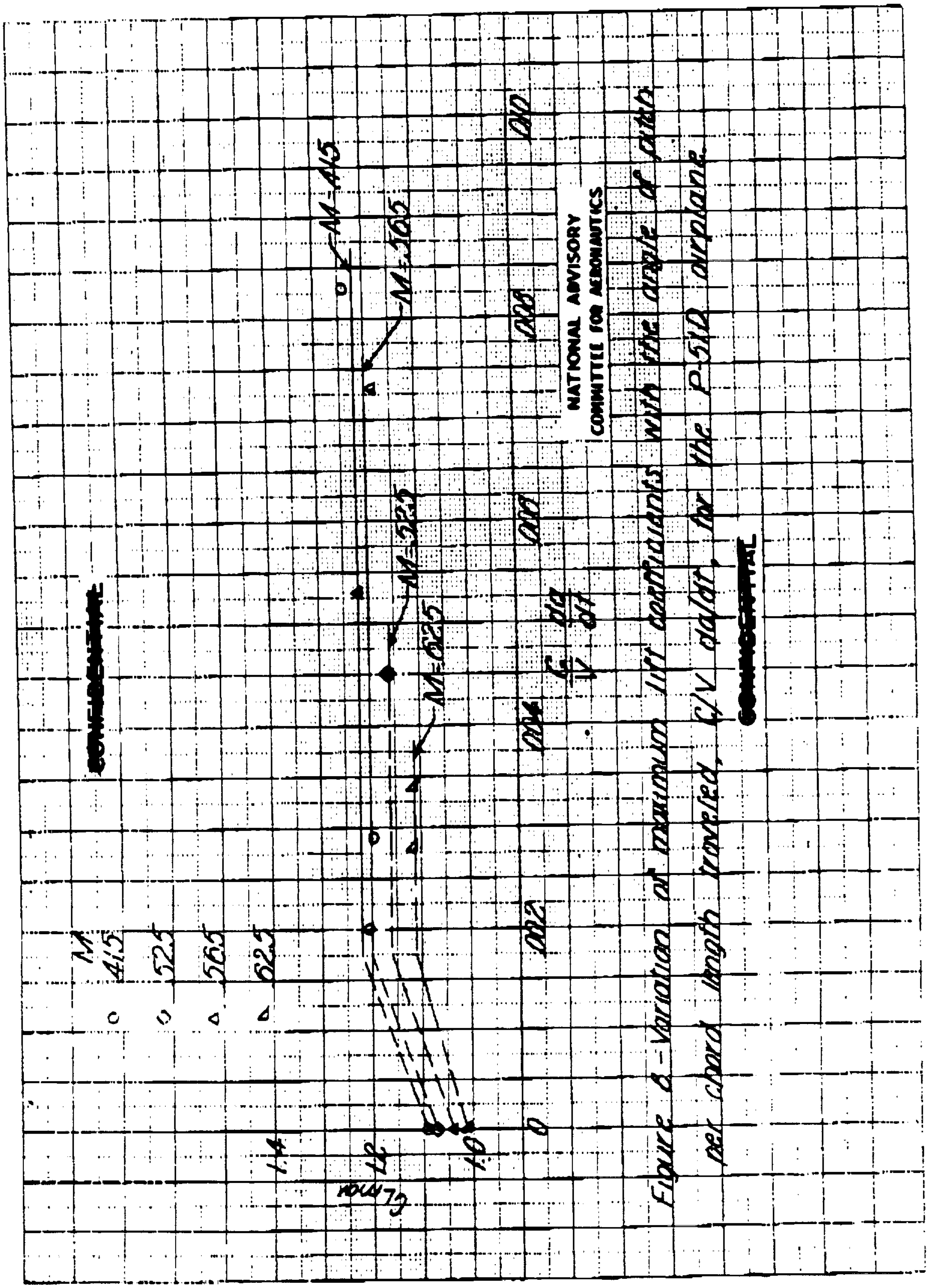


Figure 8.- Variation of maximum lift coefficients with the angle of pitch per chord length traveled, c/V dα/dt, for the P-51D airplane.